CHAPTER 3
MISCELLANEOUS GUIDANCE (MG)

AC 27 MG 8. (Amendment 27-30) SUBSTANTIATION OF COMPOSITE ROTORCRAFT STRUCTURE

a. Reference FAR Sections §§ 27.305, .307, .571, .603, .605, .609, .610, .611, .613, .629, .923, .927, .931, .1529 and Appendix A.

b. Purpose. These substantiation procedures provide a more specialized supplement to the general procedures outlined by AC 20-107A, “Composite Aircraft Structure.” These procedures address substantiation requirements for composite material system constituents, composite material systems, and composite structures common to rotorcraft. A uniform approach to composite structural substantiation is desirable, but it is recognized that in a continually developing technical area which has diverse industrial roots, both in aerospace and in other industries, some variations and deviations from the procedures described herein will be both necessary and acceptable. Significant deviations from this material should be coordinated in advance with the Rotorcraft Directorate.

c. Special Considerations. Since rotorcraft structure is configured uniquely and is inherently subjected to severe cyclic stresses, special consideration is required for the substantiation of all rotorcraft structure, including composites. This special consideration is necessary to ensure that the level of safety intended by the current regulations is attained during the type certification process for all structure with special emphasis on composite structure because of its unique structural characteristics, manufacturing quality and operational considerations, and failure mechanisms.

d. Background.

(1) Historically, rotorcraft have required unique, conservative structural substantiation because of unique configuration effects, unique loading considerations, severe fatigue spectrum effects, and the specialized comprehensive fatigue testing required by these effects. Rotorcraft structural static strength substantiation for both metal and composite structure is essentially identical to that for fixed wing structure once basic loads have been determined. However, rotorcraft structural fatigue substantiation for metals is significantly different from fixed wing fatigue substantiation. Since AC 20-107A, as developed, applies to both fixed wing aircraft and rotorcraft; it, of necessity, was finalized in a broad generic form. Accordingly, a need to supplement AC 20-107A for rotorcraft was recognized during type certification programs. One significant difference in traditional rotorcraft fatigue substantiation programs and fixed wing fatigue programs is the use of multiple full-scale specimen fatigue tests for rotorcraft programs rather than just one full-scale specimen test. Also, constant amplitude, accelerated load tests are typically used rather than spectrum tests because of the high frequency loads common to rotorcraft operations. These rotorcraft fatigue

tests have traditionally involved the generation of stress versus life or cycle (S-N) curves for each critical part (most of which are subjected to the cyclic loading of the main or tail rotor system) using a monotonic (sinusoidal) fatigue spectrum based on maximum and minimum service stress values. Unless configuration differences or flight usage data dictate otherwise, the monotonic fatigue spectrum's period is typically based on six ground-air-ground (GAG) cycles for each flight hour of operation. The S-N curves for the substantiation of each detailed part are typically generated by plotting a curved line through three data points (reference AC 29-2C, paragraph AC 29 MG 11), "Fatigue Evaluation of Transport Category Rotorcraft Structure (Including Flaw Tolerance)"). The three data points selected are a short specimen life (low cycle fatigue), an intermediate specimen life and a long specimen life (high cycle fatigue). Each raw data point is generated by monotonically fatigue testing at least two full-scale specimens (parts) to failure or run out for each data point on the S-N curve. The raw data point values are then reduced by an acceptable statistical method to a single value for plotting to ensure proper reliability of the associated S-N curve. Order 8110.9, "Handbook on Vibration Substantiation and Fatigue Evaluation of Helicopter and Other Power Transmission Systems" and paragraph AC 29 MG 11 of AC 29-2C contain comprehensive discussions of the S-N curve generation process. The rotorcraft S-N curve process contrasts sharply with the fixed wing process of using a single full-scale fatigue article (usually an entire wing or airframe, which constitutes a single full-scale assembly data point), generic material or full-scale assembly S-N data (e.g., MIL-HDBK-5 for metals, MIL-HDBK-17 for composites, or AFS-120-73-2 for full-scale assemblies), a non-monotonic spectrum and relatively large scatter factors to verify or determine the design fatigue life of the full-scale airplane.

(2) Also, rotorcraft have employed and mass produced composite designs in primary structure (typically main and tail rotor blades) since the early 1950's. This was 10 or more years before composites were type certificated for primary fixed-wing structure in either military or civil aircraft applications (with some notable limited production exceptions, such as the Windecker fixed wing aircraft). In any case, the early 1950 period was well before a clear, detailed understanding of composite structural behavior (especially in the areas of macroscopic and microscopic failure mechanisms and modes) was relatively common and readily available in a usable format for the average engineer working in this field. It also predated the initial issuance of AC 20-107. Currently, much composite design information is proprietary, either to government, industry or both, and many data gathering methods have not been completely standardized. Consequently, a significant variation from laboratory to laboratory in material property value determination methods and results can exist. The early rotor blade designs (as well as current designs) are by nature relatively low strain, tension structure designs. Also, by nature, these designs are not damage or flaw critical. Thus by circumstance as much as design, early composite rotor blade and other composite rotorcraft designs incorporated an acceptable fatigue tolerance level of safety. In the 1980's, more test data, analytical knowledge, and analytical methodology became available to more completely substantiate a composite design. Current FAR's 27 and 29 contain many sections (reference paragraph a.) to be considered in substantiating composite rotorcraft structure, but this advisory material is needed to

supplement the general guidance of AC 20-107A by providing specific rotorcraft guidance for obtaining consistent compliance with FAR sections applicable to rotorcraft.

e. Definitions. The following basic definitions are provided as a convenient reading reference. MIL-HDBK-17, and other sources, contain more complete glossaries of definitions.

(1) AUTOCLAVE. A closed apparatus usually equipped with variable conditions of vacuum, pressure and temperature. Used for bonding, compressing or curing materials.

(2) ALLOWABLES. Both A- basis and B- basis values statistically derived and used for a particular composite design.

(3) BALANCED LAMINATE. A composite laminate in which all laminae at angles other than 0° occur only in ± pairs (not necessarily adjacent).

(4) A-BASIS ALLOWABLE. The “A” mechanical property value is the value above which at least 99 percent of the population of values is expected to fall, with a confidence of 95 percent.

(5) B-BASIS ALLOWABLE. The “B” mechanical property value is the value above which at least 90 percent of the population of values is expected to fall, with a confidence of 95 percent.

(6) BOND. The adhesion of one surface to another, with or without the use of an adhesive as a bonding agent.

(7) COCURE. The process of curing several different materials in a single step. Examples include the curing of various compatible resin system pre-pregs, using the same cure cycle, to produce hybrid composite structure or the curing of compatible composite materials and structural adhesives, using the same cure cycle, to produce sandwich structure or skins with integrally molded fittings.

(8) CURE. To change the properties of a thermosetting resin irreversibly by chemical reaction; i.e., condensation, ring closure, or addition. Cure may be accomplished by addition of curing (crosslinking) agents, with or without catalyst, and with or without heat.

(9) DELAMINATION. The separation of the layers of material in a laminate.

(10) DISBOND. A lack of proper adhesion in a bonded joint. This may be local or may cover a majority of the bond area. It may occur at any time in the cure or subsequent life of the bond area and may arise from a wide variety of causes.

(11) FIBER. A single homogeneous strand of material, essentially one--dimensional in the macro-behavior sense, used as a principal constituent in advanced composites because of its high axial strength and modulus.

(12) FIBER VOLUME. The volume of fiber present in the composite. This is usually expressed as a percentage volume fraction or weight fraction of the composite.

(13) FILL. The 90° yarns in a fabric, also called the woof or weft.

(14) GLASS TRANSITION. The reversible change in an amorphous polymer or in amorphous regions of a partially crystalline polymer from (or to) a viscous or rubbery condition to (or from) a hard and relatively brittle one.

(15) GLASS TRANSITION TEMPERATURE. The approximate midpoint of the temperature range over which the glass transition takes place.

(16) HYBRID. Any mixture of fiber types (i.e., graphite and glass).

(17) IMPREGNATE. An application of resin onto fibers or fabrics by several processes: hot melt, solution coat, or hand lay-up.

(18) LAMINA. A single ply or layer in a laminate in which all fibers have the same fiber orientation.

(19) LAMINATE. A product made by bonding together two or more layers or laminae of material or materials.

(20) LOW STRAIN LEVEL. As used herein, is defined as a principal, elastic axial gross strain level, that for a given composite structure provides for no flaw growth and thus provides damage tolerance of the maximum defects allowed during the certification process using the approved design fatigue spectrum.

(21) MATERIAL SYSTEM CONSTITUENT. A single constituent (ingredient) chosen for a material system (e.g., a fiber, a resin).

(22) MATERIAL SYSTEM. The combination of single constituents chosen (e.g., fiber and resin).

(23) MATRIX. The essentially homogeneous material in which the fibers or filaments of a composite are embedded. The resins used in most aircraft structure are thermoset polymers.

(24) MAXIMUM STRUCTURAL TEMPERATURE. The temperature of a part, panel or structural element due to service parameters such as incident heat fluxes, temperature, and air flow at the time of occurrence of any critical load case, (i.e., each

critical load case has an associated maximum structural temperature). This term is synonymous with the term “maximum panel temperature.”

(25) <u>POROSITY</u>. A condition of trapped pockets of air, gas, or void within a solid materials, usually expressed as a percentage of the total nonsolid volume to the total volume (solid + nonsolid) of a unit quantity of material.

(26) <u>PRE-PREG, PREIMPREGNATED</u>. A combination of mat, fabric, nonwoven material, tape, or roving already impregnated with resin, usually partially cured, and ready for manufacturing use in a final product which will involve complete curing. Prepreg is usually drapable, tacky and can be easily handled.

(27) <u>RESIN</u>. An organic material with indefinite and usually high molecular weight and no sharp melting point.

(28) <u>RESIN CONTENT</u>. The amount of matrix present in a composite either by percent weight or percent volume.

(29) <u>SECONDARY BONDING</u>. The joining together, by the process of adhesive bonding, of two or more already-cured composite parts, during which the only chemical or thermal reaction occurring is the curing of the adhesive itself. The joining together of one already-cured composite part to an uncured composite part, through the curing of the resin of the uncured part, is also considered for the purposes of this advisory circular to be a secondary bonding operation (See COCURING).

(30) <u>SHELF LIFE</u>. The length of time a material, substance, product, or reagent can be stored under specified environmental conditions and continue to meet all applicable specification requirements and/or remain suitable for its intended function.

(31) <u>STRAIN LEVEL</u>. As used herein, is defined as the principal axial gross strain of a part or component due to the principal load or combinations of loads applied by a critical load case considered in the structural analysis (e.g., tension, bending, bending-tension, etc.). Strain level is generally measured in thousandths of an inch per unit inch of part or microinches/per inch (e.g., .003 in/in equals 3000 microinches/inch).

(32) <u>SYMMETRICAL LAMINATE</u>. A composite laminate in which the ply orientation is symmetrical about the laminate midplane.

(33) <u>TAPE</u>. Hot melt impregnated fibers forming unidirectional pre-preg.

(34) <u>THERMOPLASTIC</u>. A plastic that repeatedly can be softened by heating and hardened by cooling through a temperature range characteristic of the plastic, and when in the softened stage, can be shaped by flow into articles by molding or extrusion.

(35) THERMOSET (OR CHEMSET). A plastic that once set or molded cannot be re-set or remolded because it undergoes a chemical change; (i.e., it is substantially infusible and insoluble after having been cured by heat or other means).

(36) WARP. Yarns extended along the length of the fabric (in the 0° direction) and being crossed by the fill yarns (90° fibers).

(37) WORK LIFE. The period during which a compound, after mixing with a catalyst, solvent, or other compounding constituents, remains suitable for its intended use.

f. RELATED REGULATORY AND GUIDANCE MATERIAL.

Document	Title
(1) AC 20-95	"Fatigue Evaluation of Rotorcraft Structure"
(2) AC 20-107	"Composite Aircraft Structure"
(3) AC 21-26	"Quality Control for the Manufacture of Composite Materials"
(4) MIL-HDBK-17	"Polymer Matrix Composites Volume 1: Guidelines"

g. PROCEDURES FOR SUBSTANTIATION OF ROTORCRAFT COMPOSITE STRUCTURE. The composite structures evaluation has been divided into eight basic regulatory areas to provide focus on relevant regulatory requirements. These eight areas are: (1) fabrication requirements; (2) basic constituent, pre-preg and laminate material acceptance requirements and material property determination requirements; (3) protection of structure; (4) lightning protection; (5) static strength evaluation; (6) damage tolerance and fatigue evaluation; (7) dynamic loading and response evaluation; and (8) special repair and continued airworthiness requirements. Original as well as alternate or substitute material system constituents (e.g., fibers, resins, etc.), material systems (combinations of constituents and adhesives), and composite designs (laminates, cocured assemblies, bonded assemblies, etc.) should be qualified in accordance with the methodology presented in the following paragraphs. Each regulatory area will be addressed in turn. It is important to remember that proper certification of a composite structure is an incremental, building block process which involves phased FAA/AUTHORITY involvement and incremental approval in each of the various areas outlined herein. It is strongly recommended that a FAA/AUTHORITY certification team approach be used for composite structural substantiation. The team should consist of FAA/AUTHORITY engineering, the MIDO inspector(s), the associated Designated Engineering Representatives (DER's), the associated Designated Manufacturing Inspection Representatives (DMIR's), and cognizant members of the applicant's organization. Personnel who are composites specialists (or are otherwise knowledgeable in the subject) should be primary team member candidates. Once selected, it is recommended that team meetings be held periodically (possibly in conjunction with type boards) during certification to ensure the building block certification process is accomplished as intended.

(1) The <u>first</u> area is the <u>fabrication requirements</u> of § 27.605:

(i) The quality control system should be developed considering the critical engineering, manufacturing, and quality requirements and a guidance standard such as AC 21-26, "Quality Control For the Manufacture of Composite Materials." This ensures that all special engineering, or manufacturing quality instructions for composites are presented, evaluated, documented, and approved, using drawings, process and manufacturing specifications, standards, or other equivalent means. This should be one of the early phases of a composite structure certification program, since this represents a major building block for sequential substantiation work.

(ii) Specific allowable defect limits on, for example, fiber waviness, warp defects, fill defects, porosity, hole edge effects, edge defects, resin content, large area debonds, and delaminations, etc., for a particular material system component, laminate design, detailed part, or assembly should be jointly established by engineering, manufacturing, and quality and the associated inspection programs for defect detection created, validated, and approved. Each critical engineering design should consider the worse-case effects of the manufacturing process (maximum waviness, disbonds, delaminations, and other critical defects) allowed by the reliability limitations of the approved inspection program.

(iii) If bonds or bond lines such as those typical of rotorcraft rotor blade structure are used, special inspection methods, special fabrication methods or other approved verification methods (e.g., engineering proof tests, reference paragraph g(5)) should be provided to detect and limit disbonds or understrength bonds.

(iv) Structurally critical composite construction fabrication process and procurement specifications, for fabricating reproducible and reliable structure, must be provided and FAA/AUTHORITY approved early during the certification process and should, as a minimum, cover the following:

(A) <u>Vendor and Qualified Parts List (QPL) Control</u>. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both the manufacturing and inspection district office (MIDO) and FAA/AUTHORITY engineering) at any time, that their quality control systems ensure on a continuous basis, that only qualified suppliers provide the basic material constituents or material systems (e.g., pre-pregs) that meet approved material specifications. Recommended guidelines for qualification of alternate material systems and suppliers are contained in MIL-HDBK-17B, Volume I, Section 2.3.2. These methods can also be used, periodically for qualification status renewals of existing material systems and suppliers.

(B) Receiving Inspection and In Process Inspection. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their receiving and in-process quality control systems provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such that the necessary statistical reliability and confidence levels for the items being inspected (that are specified by engineering) are continuously maintained. This will require periodic standard inspections and engineering characterization tests on basic constituent and material system samples which should be conducted, as a minimum, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY-witnessed.

(C) Material System Component Storage and Handling. Applicants should be able to demonstrate to FAA/AUTHORITY certification team members (both MIDO and engineering), at any time, that their composite material system (or constituent) storage and handling procedures and specifications provide products which continuously meet approved material and process specifications. Quality systems should be designed with appropriate checks and balances, such that the necessary statistical reliability and confidence levels for the items being inspected (which are specified by engineering) are continuously maintained. This should require, as a minimum, periodic inspections to ensure that proper records are kept on critical parameters (e.g., room temperature “bench” exposure, shelf life, etc.) and that periodic basic constituent and material system characterization tests are conducted, on a batch-to-batch basis. The periodic testing necessary to maintain the quality standard should be conducted by the applicants on conformed samples and should be FAA/AUTHORITY-witnessed.

(D) Statistical Validation Level. It is necessary to maintain the minimum required statistical validation level of the quality control system (which should be specified for each critical item or constituent by the approved quality and engineering specifications). The statistical validation level should be defined and approved early in certification. Also, approval and proper usage should be continuously maintained during the entire procurement and manufacturing cycles.

(v) Alternate fabrication and process techniques should be approved and should comply with § 27.605. Any alternate techniques should provide at least the same level of quality and safety as the original technique. Any changes should be presented and FAA/AUTHORITY-approved well in advance of the change’s production effectivity.

(2) The second area is the basic raw constituent, pre-preg, and laminate material acceptance requirements and material property determination requirements of §§ 27.603 and 27.613. These criteria require application of the critical environmental limits such as temperature, humidity, and exposure to aircraft fluids (such as fuel, oils,

and hydraulic fluids), to determine their effect on the performance of each composite material system. Temperature and humidity effects are commonly considered by coupon and component tests utilizing preconditioned test specimens for each material system selected. Material “A” & “B” basis allowable strength values and other basic material properties (based on MIL-HDBK-17, or equivalent) are typically determined by small scale tests, such as coupon tests, for use in certification work. In the case of composites, determination of these basic constituent and material system properties will almost invariably involve the submittal, acceptance and use of company standards. This is currently necessary because MIL-HDBK-17 has not completed development of “B” basis allowables for inclusion in the handbook. Also, test methods vary somewhat from manufacturer to manufacturer; therefore, individual company results will exhibit some scatter in final material property values. Any company standard which is approved and used should meet or exceed related MIL-HDBK-17 requirements. Material structural acceptance criteria and property determination should, as a minimum, include the following:

(i) Property characterization requirements of all material systems (e.g., pre-pregs, adhesives, etc.) and constituents (e.g., fibers, resins, etc.) should be identified, documented, and approved. These requirements, once approved, should be placed in all appropriate procedures and specifications (such as those in g(1) above).

(ii) Moisture conditioning of test coupons, parts, subassemblies, or assemblies should be accomplished in accordance with MIL-HDBK-17, other similar approved methods or per FAA/AUTHORITY approved programs.

(iii) The maximum and minimum temperatures expected in service (as derived from test measurements, thermal analyses on panels and other parts, experience, or a combination) should be determined and accounted for in static and fatigue strength (including damage tolerance) substantiation programs considering associated humidity induced effects.

(iv) The glass transition temperature, Tg, is an important characteristic parameter of amorphous polymers, such as epoxies. It is the temperature below which the polymer behaves like a “glassy” solid and above which it behaves like a “rubbery” solid, i.e., it is the temperature at which there is a very rapid change in physical properties. In actuality, the change from a hard polymeric material to a rubbery material takes place over a narrow temperature range. A composite material will experience a drastic reduction in matrix controlled mechanical material properties when loaded in this temperature range. Since the resin (matrix) is the critical structural constituent in a composite and since Tg exceedance is critical to structural integrity; Tg determination is necessary. The Tg margin methodology of MIL-HDBK-17, Section 2.2.2.1, should be implemented, i.e., the wet glass transition temperature (Tg) should be 50° F higher than the maximum structural temperature (see definition). For any type of resin or adhesive, an acceptable temperature margin using MIL-HDBK-17 techniques (e.g., consideration of limited high temperature excursions) or equivalent methodologies based on tests and/or experience should be established and approved early in the certification process.

In no case should structural strength be degraded below limit load capability on a maximum world wide high temperature day.

(v) Local design values should be established by analysis and characterization tests and approved for specific structural configurations (point designs) which include the effects of stress risers (e.g., holes, notches, etc.) and structural discontinuities (e.g., joints, splices, etc.). Proper determination of these values for full-scale design and test should be considered one of the most critical building blocks in substantiating and evaluating a composite structure. These transitional load transfer areas typically produce the highest stresses (and strains) and serve as the nucleation sites for many of the failures (including those due to the relatively low interlaminar strength of composites) that occur in service in a full-scale part or assembly. Small scales tests (such as coupon, element, and subcomponent tests), or equivalent approved testing programs, and analytical techniques should be carefully designed, prepared, and approved to evaluate potential “hot spots” and provide accurate simulations and representations of full-scale article stresses and strains in the critical transition areas. Proper certification work in this area will ensure initial safety and continued airworthiness in full-scale production articles.

(vi) The design strain level for each major component and material system should be established and approved such that specified impact damage considerations are defined and properly limited. The effects of the approved strain levels should be established for each composite material using small scale characterization tests and the results should be used to establish or verify the maximum allowable design strain level for each full-scale article. The maximum allowable design strain values selected should also take into account the reliability and confidence levels established for the relevant portions of the quality control system. This methodology is necessary because the amount and size of flaws in the production article may restrict the allowable level of design strain. In a no-flaw-growth design, the maximum specified impact damage and manufacturing flaw size at the most critical location on the part will be a major factor in determining the maximum allowable elastic strain. This design approach is currently selected for nearly all civil and most military applications; since, under normal conditions, only visual inspections are required in the field (unless unusual external damage circumstances such as a hail storm occur) to maintain the initial level of airworthiness (safety). However, many military applications because of their demanding missions, employ scheduled field non-destructive inspection (NDI) maintenance, (such as comparative ultrasonics) to ensure that flaw growth either does not occur, is controlled by approved structural repair, or by replacement of affected parts. To date, civil applications have not been presented that desire a flaw growth, phased NDI approach. Therefore, selection of the full-scale article’s design strain limit based on small scale tests for a no flaw growth design is seen to be extremely important.

(vii) Composite and adhesive properties should be determined such that detrimental structural creep does not occur under the sustained loads and environments expected in service. Small scale characterization tests (such as coupon, element, and subcomponent tests) and analysis, which verify and establish the full-scale design criteria and parameters necessary to ensure that detrimental structural creep in full-scale structure does not occur in service, should be conducted early in certification and should be FAA/AUTHORITY-approved.

(viii) Material allowable strength values for full-scale design and testing should be developed using the coupon procedures presented in MIL-HDBK-17 or equivalent. At least three batches of material samples should be used in material allowable strength testing. Company standards should be prepared, evaluated and FAA/AUTHORITY approved early in certification (as part of the building block process), that reflect the material property determination considerations recommended in MIL-HDBK-17 on a equal to or better than basis.

(3) The <u>third</u> area is the <u>protection of structure</u> as required by § 27.609. Protection against thermal and humidity effects and other environmental effects (e.g., weathering, abrasion, fretting, hail, ultraviolet radiation, chemical effects, accidental damage, etc.) should be provided, or the structural substantiation should consider the results of those effects for which total protection is impractical. Determination and approval of worst-case or most conservative operating limits, and damage scenarios should be accomplished. Appropriate flammability and fire resistance requirements should also be considered in selecting and protecting composite structure. Usually a hazard analysis is conducted early in certification which identifies the various threats and threat levels for which protection must be provided. This data is then used to construct and submit for approval the methods-of-compliance necessary to provide proper structural protection.

(4) The <u>fourth</u> area is the <u>lightning protection</u> requirements of § 27.610. Protection should be provided and substantiated in accordance with analysis and with tests such as those of AC 20-53A and FAA Report DOT/FAA/CT-86/8. For composite structure projects involving rotorcraft certified to earlier certification bases (which do not automatically include the lightning protection requirements of § 27.610), these requirements should be imposed as special conditions. The design should be reviewed early in certification to ensure proper protection is present. The substantiation test program should also be established, reviewed and approved early to ensure proper substantiation.

(5) The <u>fifth</u> area is the <u>static strength evaluation</u> requirements of §§ 27.305 and 27.307 for composite structure. Only conservative proven methods of static analysis and failure criteria should be employed. The material stress-strain curve should be clearly established, at least through the ultimate design load, for each composite design. Composite structure should be statistically demonstrated, incrementally, through a program of analysis, coupon tests, minor component ultimate load tests and major component ultimate load tests. The static strength substantiation program should consider all critical loading conditions for all critical structure including residual strength and stiffness requirements after a predetermined length of service, e.g., end of life (EOL) (which takes into account damage and other degradation due to

the service period). Analytical reports and tests should consider all possible failure modes and should include the critical, allowable effects of:

(i) Environment (reference g(2) and (3) of this AC paragraph.)

(ii) Service Life (residual limit strength and stiffness demonstration.)

(iii) Load path loss (fail-safe analysis and limit strength demonstration.)

(iv) The standard fabrication process and its variability.

(v) Impact damage expected during service up to the established threshold of detectability of the field inspection methods to be employed.

(vi) Point design and structural discontinuity considerations (e.g., stress risers, joints, etc.)

(vii) Unless the ultimate strength of each critical bonded joint can be reliably substantiated in production by NDI techniques (or other equivalent, approved techniques), then limit load capability is guaranteed by either of the following or a combination thereof:

(A) The maximum disbond of each critical bonded joint which will carry limit load is established by test, analysis, or both. Disbonds greater than these values are typically prevented by design features.

(B) Each critical bonded joint on each production article should be proof tested to the critical limit load.

(viii) For static strength analysis laminae and laminate “A” and “B” basis allowables (determined in accordance with g(2) of this AC paragraph) should be used subject to the following conditions unless lower material properties are required by point design considerations (e.g., stress risers, joints, etc.) stiffness requirements (e.g., flutter or vibration margins), fatigue strength (including damage tolerance), or other overriding considerations.

(A) When applied loads are distributed through a single load path or single member within an assembly, the failure of which would result in the loss of the structural integrity of the component involved or inability of the rotorcraft structure to carry limit load, the part should be designed, analyzed, and tested using “A” basis allowables.

(B) Redundant (fail-safe) structures in which the failure of individual elements would result in applied loads being safely redistributed to other load carrying members without exceeding the limit load capability of the rotorcraft structure may be designed, analyzed, and tested using “B” basis allowables.

(6) The sixth area is the fatigue evaluation requirements of § 27.571. The fatigue evaluation method for the rotorcraft being certified should consider damage tolerance in accordance with AC 20-107A.

(i) The safe-life method for composite structure as defined in AC 20-107A is a flaw tolerant safe-life method (e.g., the test specimens consider inherent production flaws and impact damage (reference g(7)(ii) of this AC paragraph).

(ii) Large area disbonds, weak bonds, delaminations, or other defects should be considered in tests or be prevented or be limited by appropriate flaw tolerant special design features and by special manufacturing, maintenance, and inspection procedures. Special attention should be assigned to all pure bond lines (reference g(5) of this AC paragraph).

(iii) Non-fail-safe or partially fail-safe dynamic component structure, which may employ bond lines as the only load path, should be designed to relatively small previously approved values of elastic, ultimate strain for the material system utilized, and should be subjected to full-scale S-N curve testing. Six or more specimens are recommended, as part of the substantiation process. Where practical, flight-by-flight spectrum testing should be used.

(iv) All critical safety of flight composite structure must be designed to be flaw (damage) tolerant. Environment degradation and in-service damage critical values are typically included in the flaw tolerance evaluation. All other key factors, such as material selection, manufacturing, and quality assurance controls, and in-service inspection and maintenance, as noted previously, are also to be accounted for.

(v) The fail-safe design features of the rotor heads and blade retention systems, other critical primary composite structure, and point design features (e.g., bonded metal-to-composite joints) should be assessed and appropriate inspection programs provided to prevent catastrophic failure from flaw/damage propagation.

(vi) The method of generating S-N curves using approved raw data should be demonstrated, evaluated, and approved.

(vii) Any limited life items must be identified and placed in the Airworthiness Limitations section of the maintenance manual in accordance with § 27.571.

(viii) Load spectra, load truncation methods and all other major aspects of the fatigue evaluation are documented in test proposals and approved.

(ix) Flaw growth rates (from initial detectability to the established value for residual strength) must be previously established and closely monitored during substantiation. This data should be used to establish special phased inspections and maintenance intervals for critical structure, as required.

(7) The seventh major area is the dynamic loading and response requirements of § 27.629 for vibration and resonance frequency determination and separation for aeroelastic stability and stability margin determination for flutter critical flight structure. Critical parts, locations, excitation modes, and separations are to be identified and substantiated. This substantiation should consist of analysis supported by tests and tests which account for repeated loading effects and environment exposure effects on critical properties, such as stiffness, mass, and damping. Initial stiffness, residual stiffness, proper critical frequency design, and structural damping are provided as necessary to prevent vibration, resonance, and flutter problems.

(i) All vibration and resonance critical composite structure are identified and properly substantiated.

(ii) All flutter-critical composite structure are identified and properly substantiated. This structure must be shown by analysis to be flutter free to 1.1 V_{NE} (or any other critical operating limit, such as V_D, for a VSTOL aircraft) with the extent of damage for which residual strength and stiffness are demonstrated.

(iii) Where appropriate, crash impact dynamics considerations must be taken into account to ensure proper crash resistance and a proper level of occupant safety for an otherwise survivable impact.

(8) The eighth area is the special repair and continued airworthiness requirements of §§ 27.611, 27.1529, and FAR Part 27 Appendix A for composite structures. When repair and continued airworthiness procedures are provided in service documents (including approved sections of the maintenance manual or instructions for continued airworthiness) the resulting repairs and maintenance provisions must be shown to provide structure which continually meets the guidance of paragraphs (1) through (7) of this AC paragraph. All certification based repair and continued airworthiness standards, limits, and inspections must be clearly stated and their provisions and limitations defined and documented to ensure continued airworthiness. In general, no composite repair should be attempted which is out of scope to repairs stated in an approved Structural Repair Manual (SRM) without an engineering design approval by a qualified FAA/AUTHORITY representative (DER or staff engineer). The following minimum criteria should be met in any acceptable composite repair:

(i) The repair should be permanent.

(ii) The repair should restore the structure to the required strength and stiffness.

(iii) The repair should restore all functional requirements.

(iv) The repair should have a negligible weight penalty.

(v) The repair should be aerodynamically compatible.

(vi) The repair materials should be compatible in all essential aspects with the parent materials.

In summary, primary composite structure is an especially critical structure that requires a clearly defined, phased approval (building block) certification process. This process should involve the entire project certification team from a project's start to its finish so that proper certification is continuously and ultimately achieved. Also, in some special cases, involving new advanced state-of-the-art composite technology, an issue paper may be necessary. However, in the majority of cases (using current composite materials and design philosophy) the applicant's acknowledged use of this advisory material (as recorded in the type board minutes) should eliminate the need for a separate issue paper.